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A STUDY OF ABORT FROM A MANNED LUNAR LANDING AND RETURN TO RENDEZVOUS IN A 50-MILE ORBIT

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A STUDY OF ABORT FROM A MANNED LUNAR LANDING AND RETURN TO RENDEZVOUS IN A 50-MILE ORBIT

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SUMMARY

An investigation has been made of some of the problems associated with abort from landing and return to an orbiting vehicle in a 50-mile lunar orbit. For this study the landing module was considered capable of direct return to the orbiting vehicle from a hovering position at the lunar surface. The investigation was divided into two parts, an analytical study and a simulation study. The results of the analytical study indicate that, for an economical return to the orbiting vehicle, the landing maneuver should be chosen such that the orbiting vehicle is almost directly above the landing module at the touchdown point. This requirement places limitations on the angular travel of the landing vehicle around the moon prior to touchdown. Results of the simulation study indicate that a pilot can control the abort maneuver by using visual information.

INTRODUCTION

Recently, attention has been focused on using rendezvous to accomplish manned lunar landing missions. Both earth rendezvous and lunar rendezvous have been considered. The two methods differ in that the earth rendezvous method proposes using one vehicle for both the lunar orbit and lunar landing whereas the lunar rendezvous method proposes the use of two vehicles. The lunar rendezvous method, on which the present study is based, may be described essentially as follows: A manned vehicle approaching the moon is decelerated into a low-altitude circular orbit about the moon. From this vehicle, the landing module descends to the moon's surface. After exploration, the landing module ascends for rendezvous with the orbiting vehicle. The return vehicle is then boosted into a trajectory to the earth.

Abort considerations and techniques for the two rendezvous methods are, in general, the same for the different phases of the operation from earth launch to lunar orbit and from lunar orbit to earth reentry. (See ref. 1 for review of abort techniques for manned lunar missions.) In the lunar rendezvous method, the lunar landing phase represents a unique problem because the landing module must separate from the orbiting vehicle and, where abort during the landing maneuver is necessary, must effect a direct return to the orbiting vehicle. One desire for direct return is based on consideration of solar flares, indications being that the astronauts can avoid excessive exposure to radiation if the return to the orbiting vehicle is effected within 1 hour.

The purpose of this paper is to present the results of an analytical and a simulation study which deals with some of the abort problems associated with the landing module. Since the fuel requirements for abort are most critical just prior to touchdown, the present investigation deals only with abort from a hovering condition at the lunar surface. These results are also applicable to lunar take-off trajectories.

SYMBOLS

The British system of units is used in this study. In case conversion to metric units is desired, the following relations apply:

1 foot = 0.3048 meter

1 statute mile = 5,280 feet

h_a highest altitude of elliptical lunar trajectory, statute miles

R distance along line of sight from orbiting vehicle to landing module, statute miles or ft

t time, min

T_l longitudinal thrust (thrust provided by either the front or rear rocket of landing module), lb

T_t transverse thrust (thrust provided by either of the side rockets of landing module), lb

V elliptical transfer trajectory velocity of landing module at time of abort, ft/sec

ΔV characteristic velocity (total velocity required for the landing module to effect the abort and rendezvous maneuver), ft/sec

x_i,y_i,z_i three orthogonal components of rectangular coordinate system centered in orbiting vehicle. For the simulation study, inertial axes were used such that z_i is alined with local vertical at time of abort.

α angle subtended by line of sight between orbiting vehicle and landing module and projection of line of sight in x_i,y_i plane, deg

β angle between x_i-axis and projection of line of sight in x_i,y_i plane, deg

γ elliptical transfer trajectory flight-path angle of landing module at time of abort, deg

θ angular distance about moon traveled by orbiting vehicle from time of abort to rendezvous, deg

$\Delta\theta$ angular distance about moon by which orbiting vehicle leads landing module at time of abort, deg

A dot over a quantity denotes first derivative with respect to time.

METHOD OF ANALYSIS

This investigation is concerned with the abort problems associated with a manned module landing from a space vehicle in a 50-mile circular orbit about the moon. Figure 1 shows a schematic diagram of an abort from a hovering condition at the moon surface. The investigation was divided into two parts, an analytical study and a simulation study.

Analytical Study

In order for the landing module to abort and rendezvous with the orbiting vehicle, it is necessary for the landing module to (1) achieve a trajectory which will intercept that of the orbiting vehicle, and (2) change its velocity such that the relative velocity of the two vehicles is zero at time of interception. By assuming instantaneous velocity changes and assuming rendezvous was made the first time that the two trajectories intersected, the characteristic velocity requirements for abort were determined in the following manner. At the lunar surface, the initial velocity vector of the landing module was varied to give various elliptical trajectories that would intersect the circular orbit. For each trajectory, the relative position the orbiting vehicle would have with the landing module at time of abort and the velocity of the landing module at interception were then calculated. The characteristic velocity required for abort is the sum of the velocity required to place the landing module on the intercept trajectory and the velocity required to place the landing module in a 50-mile circular orbit at the point of interception. Standard elliptical orbital equations were used to calculate the intercept trajectories, all of which were asuumed to be in the plane of the circular orbit, for values of $\gamma \geqq 0^{\circ}$.

Simulation Study

The analog computer program used to simulate the pilot-controlled rendezvous of a space ferry vehicle with an orbiting space station, as reported in reference 2, was utilized for this study. The computer program was rescaled such that the computer simulated orbital conditions about the moon instead of orbital conditions about the earth. The purpose of the simulation study was to determine the ability of a human pilot, assuming reasonable vehicle dynamics in six degrees of freedom, to effect successfully the abort and rendezvous maneuver. The pilot was furnished with range, closure rate, and the attitude information of the vehicle. Angular motion of the orbiting station was detected by visual observation of a simulated orbiting vehicle against a simulated star background. The pilot was not presented with altitude information.

The landing module was assumed to have four rockets, one at the front, one at the rear, and one on each side. The rear rocket was provided with a throttle that had two constant thrust levels, 2,000 pounds and 200 pounds, the higher thrust level being used only during the initial phase of abort. The other three rockets had a constant thrust level of 200 pounds. Pure rotational reaction controls were assumed for attitude and were used for alining thrust in the proper direction. The assumed weight of the landing module at time of abort was 1,000 pounds.

For the initial acceleration of 2 earth g (2,000 pounds continuous thrust for the 1,000-pound module) the module required only 1.81° of travel for γ held constant at 0°, and a characteristic velocity of only 10 feet per second greater than the impulsive velocity value, to accelerate to lunar orbit velocity. The 2,000-pound thrust, therefore, can be considered to represent closely impulsive thrust conditions.

RESULTS AND DISCUSSION

Analytical Study

The results of the analytical study, shown in figures 2 to 4, are presented as characteristic velocity requirement curves for the abort and rendezvous maneuver. The boundaries that are imposed on the elliptical intercept trajectories are (1) $\gamma = 0^{\circ}$ and (2) $h_a = 50$ miles (from vertical launch to Hohmann transfer).

The characteristic velocity requirements for the landing module to abort from a hovering condition (at the moon surface) are presented in figure 2 as a function of the angle traveled by the landing module from time of abort to rendezvous. The characteristic velocity requirements are presented for several initial elliptical transfer trajectory velocities. Shown in figure 2 is a dashed curve (labeled $\Delta\theta = 0^{\circ}$) for the case where the orbiting vehicle is directly above the landing module at time of abort. Thus, when the orbiting vehicle is leading (ahead of) the landing module at time of abort, the appropriate abort conditions are above or to the right of the curve for $\Delta\theta = 0^{\circ}$.

The velocity requirements shown in figure 2 are also presented in figure 3 as a function of the angular travel of the orbiting vehicle and in figure 4 as a function of the angle between the vehicle and landing module at time of abort. For convenience, the horizontal scale of figure 3 is given both in angular travel of the orbiting vehicle and in time required for the maneuver.

The results in figures 3 and 4 show that, by the proper selection of initial velocity and flight-path angle such that the intercept trajectory is tangential at the point of rendezvous ($h_a = 50$ miles), the landing module can make an efficient direct return to the orbiting vehicle in considerably less than 1 hour.

For the Hohmann transfer ($\gamma = 0^{\circ}$; $h_a = 50$ miles) the orbiting vehicle would be leading the landing module by 5.9° at time of abort and the maneuver would require approximately 56 minutes. Figure 4 shows that for a range of initial positions of the orbiting vehicle from $\Delta\theta = -10.5^{\circ}$ to $\Delta\theta = 5.9^{\circ}$ the characteristic velocity requirements for the trajectories at $h_a = 50$ miles are 5,900 feet per second or less (5 percent < Hohmann).

As noted in the figures, the results are limited to values of characteristic velocity below 7,500 feet per second. This value of 7,500 feet per second for characteristic velocity is that capability actually being considered for the abort maneuver in a manned lunar landing mission. The results in figure 4 show that the abort maneuver can be made with this velocity capability for lag angles of the orbiting vehicle at time of abort up to 16.9°. This lag angle ($\theta = -16.9^{\circ}$) is associated with the end point of the boundary at $h_a = 50$ miles shown in figure 2 (vertical launch). Hence, limitations must be imposed on the angular travel of the landing module prior to touchdown.

As shown in figure 4, the abort and rendezvous maneuver by the present method is not possible if the orbiting vehicle is leading the landing module by more than 5.9° at time of abort since the initial flight-path angle required would be negative; this condition leads to trajectories which pass through the moon. By using other abort techniques, however, and possibly allowing more than 1 hour for the maneuver, the lead angle of the orbiting vehicle at time of abort can be increased beyond 5.9°. For example, the landing module can abort to a low-altitude circular orbit in order to catch up with the orbiting vehicle, and at the proper time make the transfer and rendezvous maneuver.

Simulation Study

For the simulation study, two initial positions for the orbiting vehicle and landing module were chosen: (1) the orbiting vehicle directly over the landing module ($\Delta\theta = 0^{\circ}$) and (2) the orbiting vehicle leading the landing module by 10°. The pilot was given the initial position and velocity of the orbiting vehicle and instructed to effect an abort and rendezvous maneuver. The pilot arbitrarily chose to use the high thrust level of the rocket until a closure rate between the two vehicles of between 900 and 1,000 feet per second was established.

Figure 5 presents typical aborts for the two initial positions of this study showing time histories of R, $\dot{R}$, $R\dot{\alpha}$, $R\dot{\beta}$, T_l, T_t, z_i, and x_i. In figure 5(a), for the case where abort was initiated when the orbiting vehicle was directly over the landing module $\Delta\theta = 0^{\circ}$, the pilot applied maximum thrust along the line of sight to initiate the abort maneuver. Maximum thrust was used until the closure rate was approximately 950 feet per second and $R\dot{\alpha}$ was zero. Thus, the pilot's task for the remainder of the run was to maintain a collision course and to effect a braking action so that the relative velocity was reduced to zero at zero range. For this run, rendezvous was made at about 30° down range from the abort position ($\theta = 30^{\circ}$) and a characteristic velocity of 7,030 feet per second was used. For the same initial conditions and angular travel, the abort and rendezvous maneuver when impulsive velocity was used, required a characteristic velocity of 5,930 feet per second. (See the dashed line in fig. 2.)

Figure 5(b) shows the time histories for the case where, at time of abort, the orbiting vehicle is leading the landing module by 10°. (It should be noted that different time scales were used for figs. 5(a) and 5(b).) To initiate the abort maneuver, the pilot chose to apply maximum thrust in a vertical direction for several seconds in order to obtain altitude before starting a chasing procedure. After obtaining altitude, the pilot pitched the vehicle over approximately 70° in order to aline the thrust vector along the line of sight. From the time histories, it can be seen that the pilot used maximum thrust the second time until the closure rate was approximately 900 feet per second. At this time, $R\dot{\alpha}$ and $R\dot{\beta}$ were not zero; therefore, transverse thrust was used to bring $R\dot{\alpha}$ and $R\dot{\beta}$ to zero. Thus, after having established a collision course, the pilot's task again was to maintain a collision course and effect a braking action so that the relative velocity was reduced to zero at zero range. Rendezvous was accomplished about 95° down range from the abort position and required 9,860 feet per second characteristic velocity.

Results of the simulation study show that a pilot can control the abort maneuver. Six piloted aborts were made for each initial position and the pilot successfully made the abort and rendezvous maneuver for all 12 runs. For the case shown in figure 5(a) where abort was initiated when the orbiting vehicle was directly above the landing module, the piloted abort maneuver required approximately 20 percent more characteristic velocity to rendezvous at $\theta = 30^{\circ}$ than was required with impulsive velocity. A comparison between the results of the simulation study and the analytical study for the case where $\Delta\theta = 10^{\circ}$ is not realistic because any intercept trajectory obtained by using instantaneous velocity changes at time of abort and to effect rendezvous for this position would pass through the lunar surface ($\gamma < 0$). Although the pilot had no altitude information during a simulation run and was given an initial position where the intercept trajectory obtained from the analytical study would pass through the lunar surface, all the piloted runs maintained a positive altitude.

CONCLUDING REMARKS

A preliminary investigation of the problems associated with abort in landing and return to a lunar orbiting vehicle in a 50-mile orbit has been made. Analytical results of the investigation indicate that the landing maneuver should be tailored such that at point of touchdown, the orbiting station should be slightly behind or, at worst, a few degrees down range from the landing module. This technique for the landing would insure an efficient return on the same orbital pass should an abort situation arise. This requirement would place limitations on the angular travel of the landing module around the moon prior to touchdown. Simulation results indicate that a pilot can control the abort maneuver by using visual information.

Langley Research Center,
National Aeronautics and Space Administration,
Langley Station, Hampton, Va., August 30, 1962.

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2. Brissenden, Roy F., Burton, Bert B., Foudriat, Edwin C., and Whitten, James B.: Analog Simulation of a Pilot-Controlled Rendezvous. NASA TN D-747, 1961.

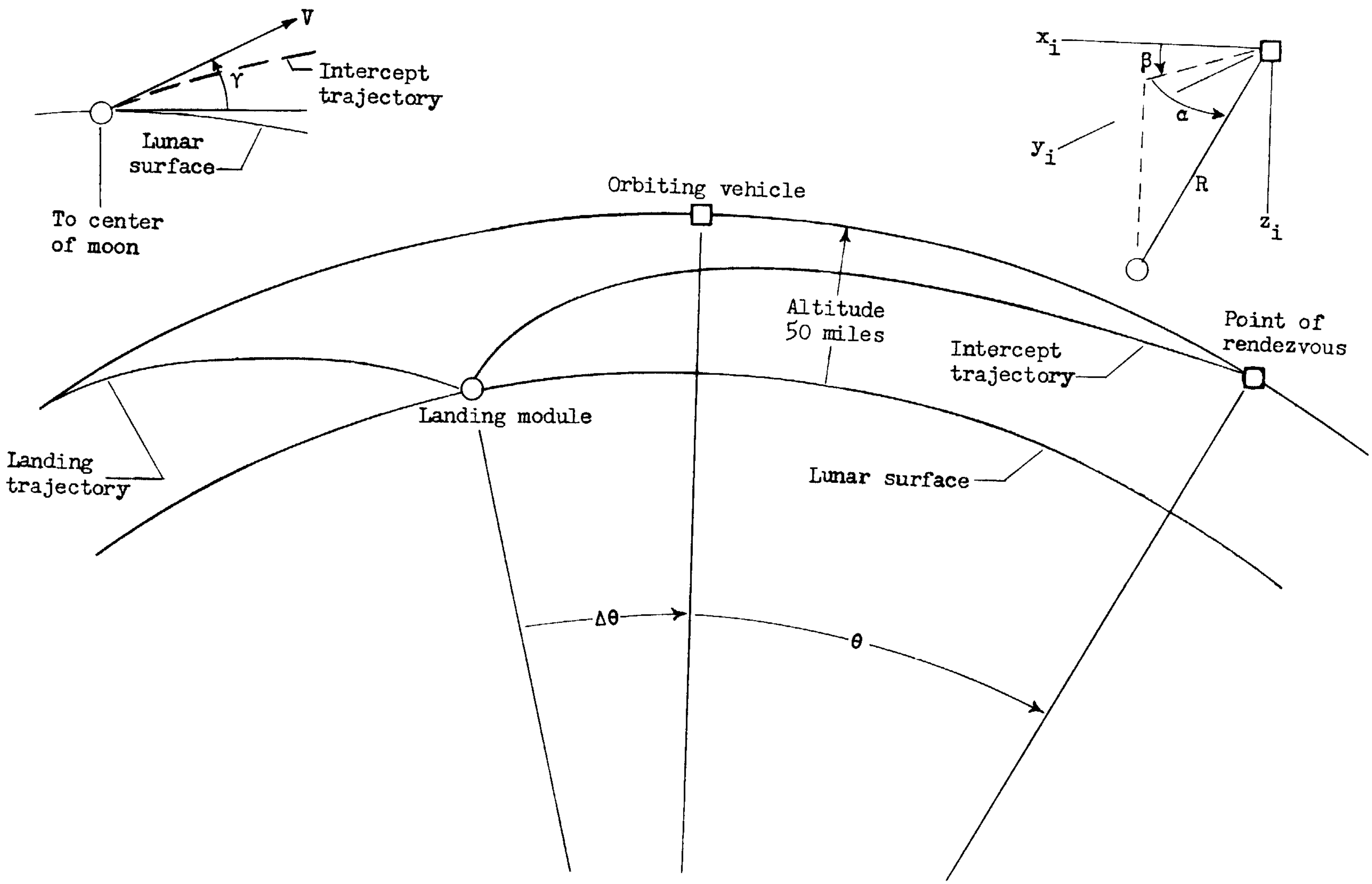


Figure 1.- Diagram showing position of landing module and orbiting vehicle at time of abort.

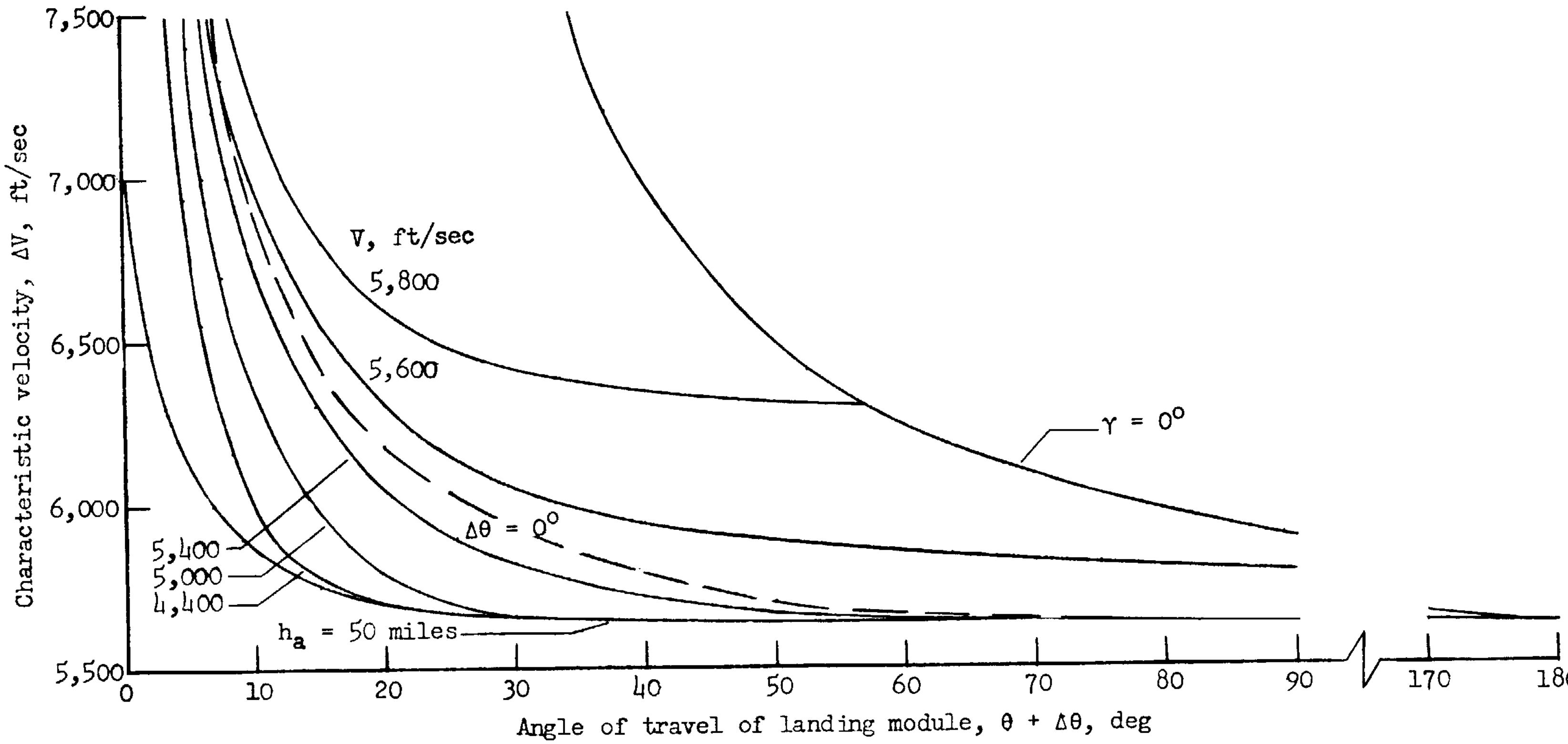


Figure 2.- Characteristic velocity requirements for the landing module to abort from a hovering condition and rendezvous with the orbiting vehicle as a function of the angle traveled by the landing module.

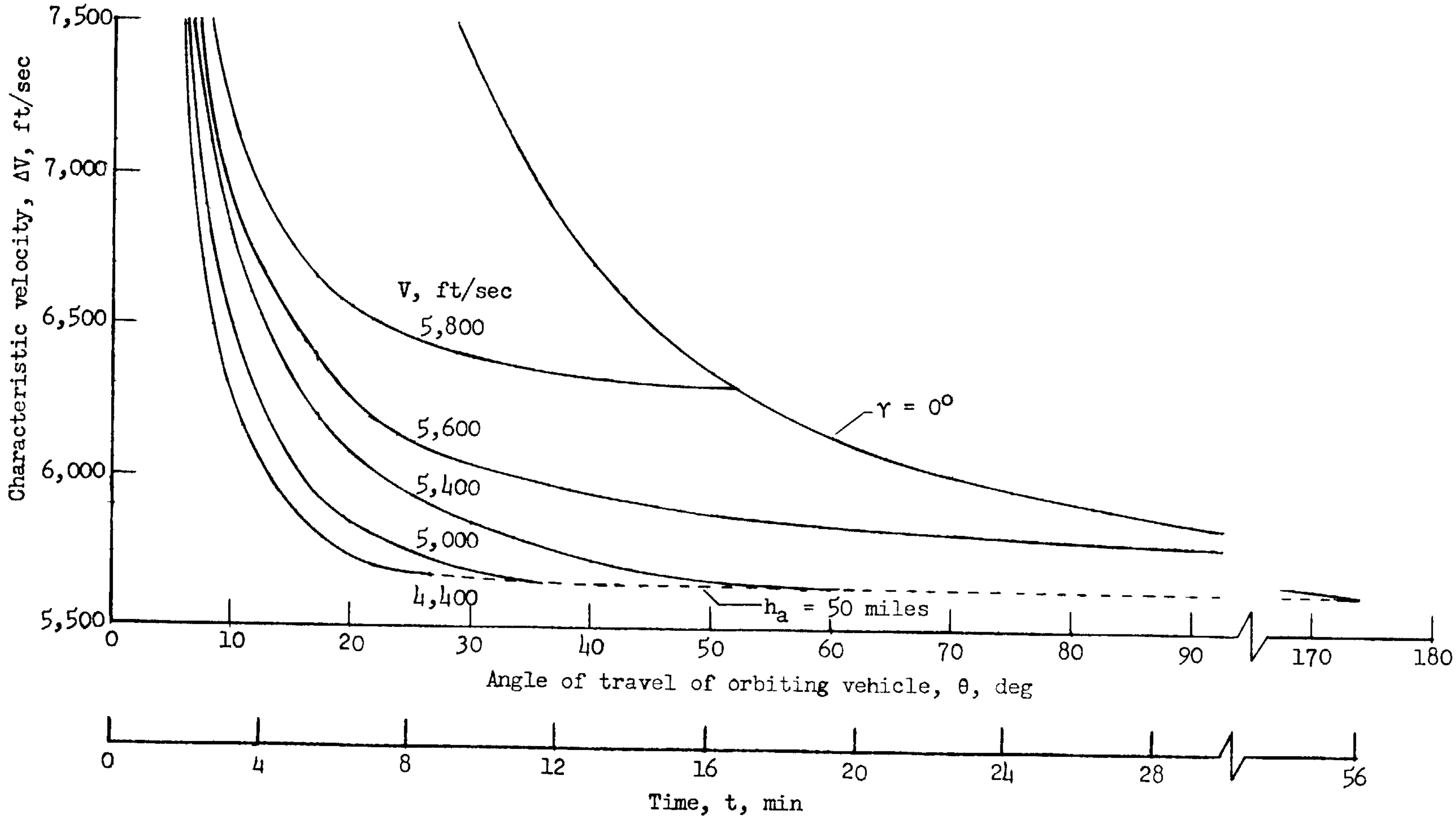


Figure 3.- Characteristic velocity requirements for the landing module to abort from a hovering condition and rendezvous with the orbiting vehicle as a function of the angle traveled by the orbiting vehicle.

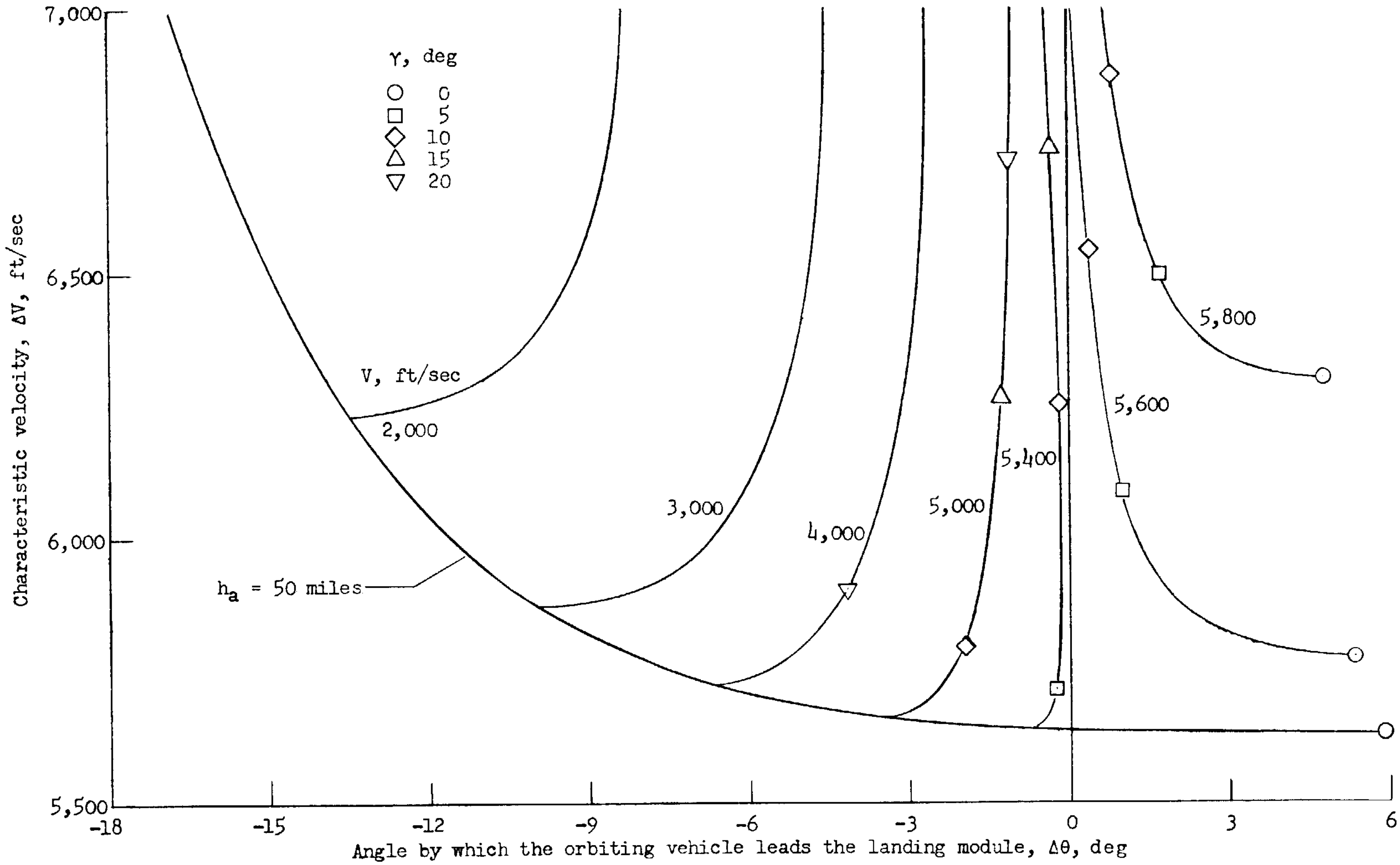
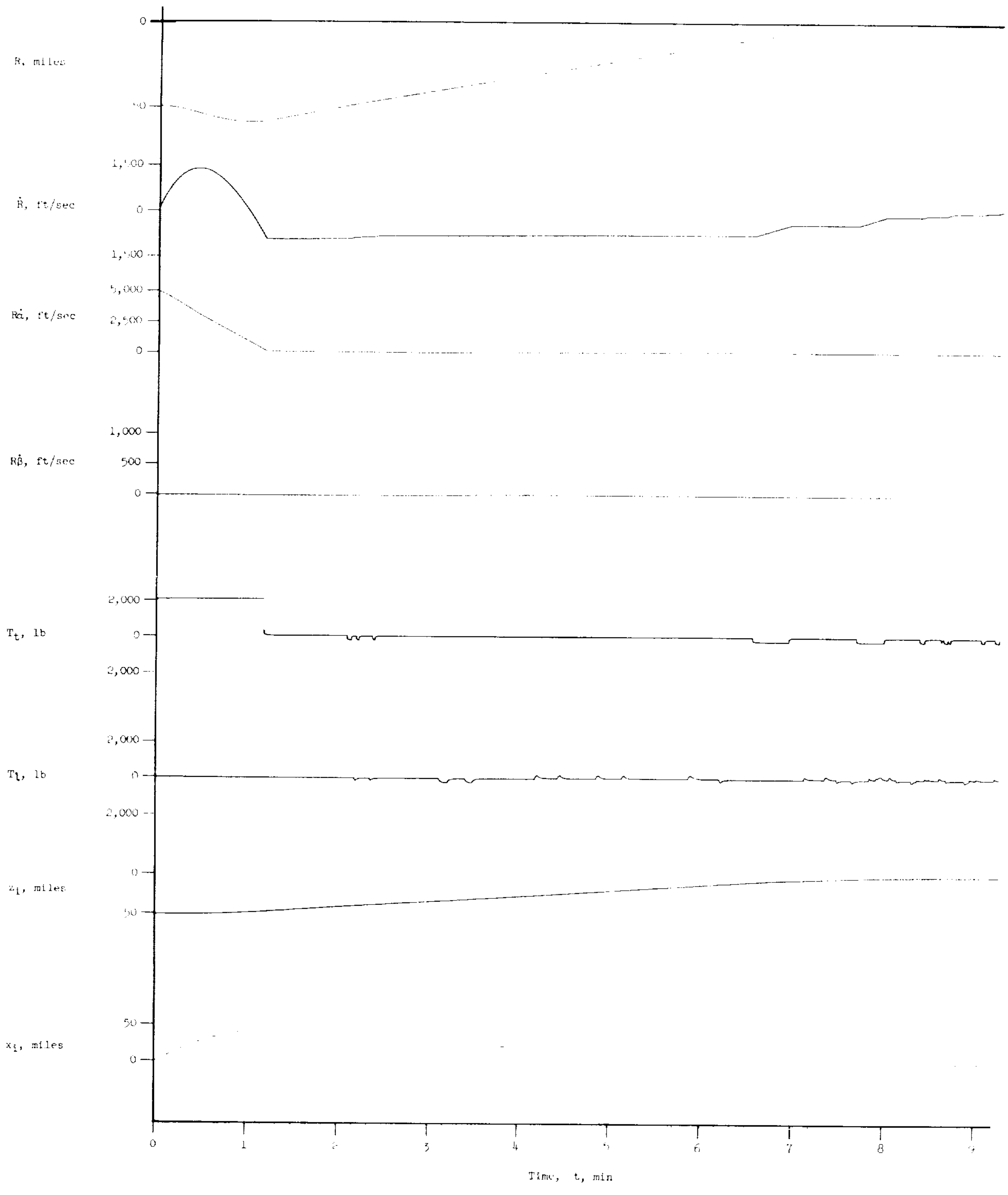
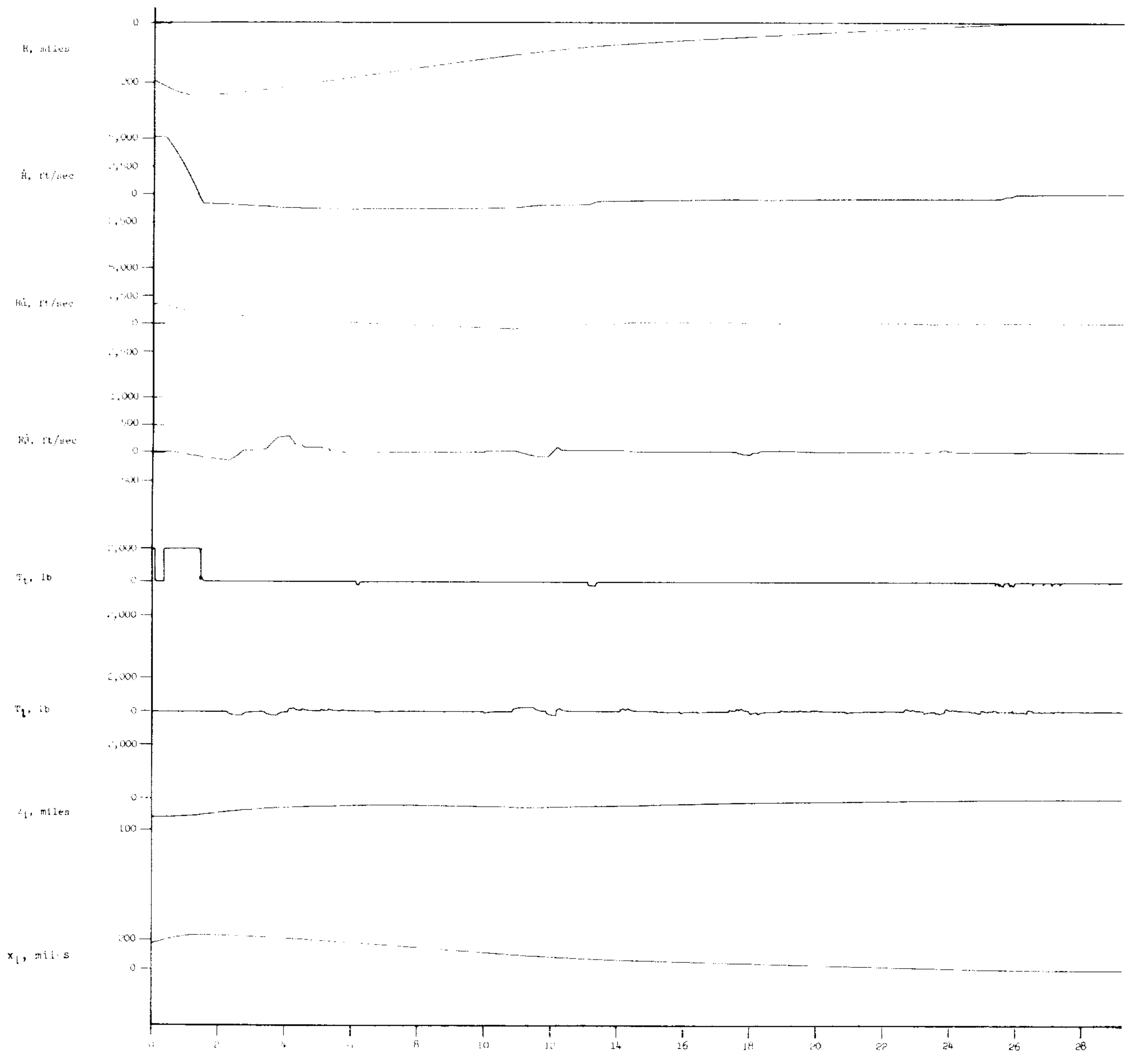


Figure 4.- Characteristic velocity requirements for the landing module to abort from a hovering condition and rendezvous with the orbiting vehicle as a function of the angle by which the orbiting vehicle leads the landing module at time of abort.



(a) $\Delta\theta = 0^{\circ}$.

Figure 5.- Time histories of typical piloted abort and rendezvous maneuvers for two initial positions of the orbiting vehicle and the landing module.



(b) $\Delta\theta = 10^{\circ}$.

Figure 5.- Concluded.